loan copy 7/3/63.

Invited Article for "SCIENCE"

N65-88847

~~X63 15760~~

28 p.

Code 2A.

(NASA TM X-50538)

6021505 NASA. Lewis Research Center, Cleveland, Ohio

ELECTRIC PROPULSION

W. E. Moeckel [1963] 24 refs Submitted for Publication

X63-15760

ELECTRIC PROPULSION

To explore our solar system adequately, we need rockets that are much better, but not necessarily much bigger, than those now being developed.

By W.E. Moeckel

In this decade, a species on planet Earth has accumulated the ability to send information-gathering vehicles to explore the nearby planets. This species is also currently engaged in a vast effort to send some of its members to the Moon. The existence of this capability and this effort illustrates an interesting attribute of this species, namely, an apparent collective urge to know and do all that it is possible to know and do. If we assume that this urge does not subside, then some members of this species must develop the means to permit, among other things, reasonably rapid traversal of progressively greater distances. This problem is largely a propulsion problem - and the propulsion problem, in turn, is largely that of providing greater amounts of useful energy with less total weight. Consequently, all of us, as members of this remarkably wide-ranging species, should be extremely interested in propulsion and energy conversion.

Propulsion is a means of producing motion, or more accurately, changes in momentum. Propulsion is also required to maintain momentum in the presence of resistive or retarding forces. These functions of a propulsion system (changing or maintaining momentum) require the expenditure of energy. If this propulsive energy must be carried by the vehicle, as is the case with most systems, then the amount of momentum change, or the duration of momentum maintenance, is determined by the form and amount of the energy carried along.

Ideally, this energy should be in a form which yields the maximum useful energy units per unit of weight.

Other factors, however, such as operating problems and cost, development problems and cost, or adequacy of existing systems, may argue against use of systems having the highest possible energy per unit weight. Thus, in the case of aircraft, the use of nuclear power would have greatly increased the energy per unit weight, but it involved numerous difficult and costly development and operational problems. Its advantages, in terms of much greater range and flight duration, were overbalanced by these disadvantages, and by the fact that kerosene-burning aircraft were already available with range, speed, and flight durations which were adequate for most applications of interest.

For space missions, if the human species were satisfied with exploring the Moon, or even settling a colony on it, there would be little incentive to develop propulsion systems with much greater energy per unit weight than the chemical rockets now in use or under development. Some increased economy might be possible, it is true, if nuclear rockets or electric rockets were to be developed, but then, on the other hand, there are numerous possible improvements in chemical rockets also, such as using the air through which the vehicle passes to enhance the useful energy, recovering and re-using the expensive launching vehicles, or developing higher energy chemicals. The net gain in using energy sources other than chemical for lunar missions alone would therefore probably be rather marginal, as it is with nuclear-propelled aircraft for terrestial missions.

It seems safe to assume, however, that mankind will not stop at the Moon, but will go on to explore first, the near planets, then the more distant planets, and eventually, perhaps, other solar systems. Why mankind should want to do this is not particularly clear. National defense or national prestige cannot be invoked indefinitely. Slogans such as "Let's beat the Russians to α-Centauri" may not have the desired wide appeal when the capability is achieved. But when the time comes that such trips are humanly possible, it will also be within human ingenuity to find excellent reasons for undertaking them. Whatever the real or apparent motivations, most people seem to understand, or feel, that it would be a serious betrayal of the human spirit, and perhaps a symptom of decline and decay, if the challenge of the vast distances of interplanetary and interstellar space is ignored.

Limitations of Chemical Rockets

Assuming, then, that mankind will wish to travel beyond the Moon, the need for propulsion systems with higher energy per unit weight than chemical rockets becomes painfully obvious. To launch the Mercury spacecraft into orbit required a launching vehicle with about 360,000 pounds of thrust, standing about the height of a 7-story building. To launch the Apollo spacecraft to the Moon will require a launching vehicle (Saturn-5) with 7-1/2 million pounds of thrust, standing about the height of a 30-story building. To send an expedition of about 7 men to explore Mars and return would require a launching vehicle with thrust of about 150 million pounds, and standing higher than a 70-story building.

Now, no one is seriously proposing to launch such a Mars expedition with a single monstrous launching vehicle. It is possible, instead, to launch parts of the interplanetary vehicle into a low orbit around the Earth, and assemble the pieces in orbit. Even this, however, becomes a chore of tremendous magnitude. It would involve 20 to 40 launchings using the huge Apollo booster (Saturn-5) and would result in an orbiting vehicle weight of 5 to 10 million pounds.

How can we reduce this weight? More than 90 percent of it is the chemical fuel and oxygen. Why is so much needed? Because the useful energy released by combustion (even with high-energy reactants, like hydrogen and oxygen) is not great enough. How can we increase this useful energy? To answer this, we must first describe what is meant by useful propulsive energy.

Useful Propulsive Energy

In space, the only way in which forward motion or momentum can be generated is to eject something rearward. This is in accordance with Newton's law that every action (such as a forward push) requires an equal and opposite reaction. For moving a vehicle on land, we can push rearward on the ground, and for sea or air propulsion we can push backward on the water or air (with propellers, or in the case of jet engines, by making the air leave the engine at higher speed than it entered). But in space we must eject material that is carried along in the vehicle, because there is nothing else substantial enough to push backwards on. The material that is ejected is quite appropriately called the <u>propellant</u> since it propels the vehicle forward.

Considering that, for space propulsion, the useful thing is the forward push (or thrust) on the vehicle, it seems clear that we can get the most useful work out of each bit of propellant by ejecting it rearward at the highest possible speed, because the faster it leaves, the more momentum it has and the more is imparted to the vehicle by reaction. Useful propulsion energy is therefore the kinetic energy of the ejected propellant, which depends only on its exit speed.

What limits the exit speed of the propellant? Primarily the energy that can be imparted to the propellant by the propulsion system. For chemical rockets, a fuel and an oxidant are burned in a combustion chamber, where the combustion products reach high temperature and pressure. The chemical energy has been transformed by combustion into heat energy. To convert this heat into the useful energy of motion (kinetic energy) of the propellant (which is, in this case, the gaseous products of combustion), the resulting gases are allowed to escape rearward through a diverging nozzle, which converts heat energy into kinetic energy. The final exit speed of the gases corresponds to a kinetic energy which can be no greater than the initial chemical energy of combustion, and is actually somewhat less. Obviously, it is best to use chemicals which produce the most heat per pound of combustion product. Hydrogen and oxygen rank very near the top in this regard, and considerable development effort on hydrogen-oxygen rockets is underway.

But, as we have seen, even hydrogen-oxygen combustion energy is not enough to produce the exit velocities needed to undertake manned interplanetary missions with reasonable propellant weight. To reduce the propellant weight we need higher propellant ejection speeds than are attainable with chemical energy. There are two alternative energy sources: nuclear energy and solar energy. How can these be used to eject propellant at higher velocities?

Propellant Heating Systems

One way is to use these energy sources to heat the propellant to higher levels than is possible with chemical energy. A major development program is, in fact, underway to produce a nuclear rocket in which hydrogen is heated by passing it through tubes in a nuclear reactor. A solar-heated hydrogen rocket is also possible, although the collector needed to concentrate enough solar energy (sun light) tends to be quite huge. In these heating methods, the energy-addition limit is the temperature achievable in the heating tubes without softening or melting them. Hydrogen is the best propellant to use with these heating schemes, because it has the largest energy when heated [illegible] top temperature. The reason for this is that, at any given heater temperature, the particles that are accelerated the most during escape through a nozzle are those that have the least mass (like hydrogen atoms).

Using these nuclear or solar heating schemes, much better performance can be attained than with chemical rockets. The energy in each bit of propellant can be quadrupled, and the exit velocity of the propellant can be about doubled, before the temperature limit of the heater is reached. This doubling of the jet velocity reduces the manned Mars exploration mission to somewhat more manageable size. Instead of launching 5 to 10 million pounds into orbit, and assembling the parts, it is now possible to launch only 1-1/2 to 3 million pounds for the same mission. This can be done with 6 to 12 Saturn-5 boosters instead of 20 to 40. So we see that the nuclear or solar hydrogen-heating rockets are extremely attractive propulsion systems. Not only are they attractive for the manned Mars mission; they would also be extremely useful in carrying the large payloads needed for establishment of a permanent lunar base.

Higher Propellant Velocities by Electric Propulsion

We have seen that the ejection speed of the propellant is limited by the energy of chemical reactions in the case of the chemical rockets, and by the temperature of the heater materials for the nuclear or solar hydrogen-heating rockets. How can we avoid these limitations, and increase the propellant ejection speed even more? The electric rocket is one answer. If an electron is removed from each atom of the propellant, then each atom will have a positive electric charge (thereby becoming a positive ion). All that is needed to accelerate an electrically-charged particle is a voltage. Therefore, by applying the proper voltage, we can accelerate the propellant particles to any exit speed that we may desire. Or, by making an electrically-conducting gas (plasma) out of the propellant, we can acclerate it with an electric current and a magnetic field, in the same way that solid conductors are moved in an ordinary electric motor. A wide variety of possibilities arise for adding as much energy as we may wish to the propellant particles.

It would seem, then, that we can reduce the amount of propellant needed for space missions to very small amounts indeed. This is quite true, but there is just one problem - the electric power-generating equipment must be carried along. For an electric powerplant, the maximum number of kilowatts generated is fixed. To increase the propellant exit speed, therefore, we must put more and more kilowatts into less and less propellant.

Thrust and Power Relations

We need three equations to show what happens. The first is the equation for thrust,

$$\text{Thrust} = \dot{W}_p \frac{v_p}{g_0} = \dot{W}_p I \qquad (1)$$

where $\dot{W}_p$ is the propellant flow rate, in units such as grams per second. The quantity v_p is the propellant ejection velocity, and g_0 is the gravitational acceleration constant (980 cm/sec^2) which relates mass to weight. The quantity I is called the specific impulse, perhaps the most common figure of merit of a rocket. It is a measure of the amount of thrust produced by a given propellant flow rate $\dot{W}_p$, and is directly proportional to propellant ejection velocity v_p $(I = v_p/g_0)$.

The second equation is for the power in the ejected propellant (also called the jet power):

$$\text{Jet Power} = 1/2\ \dot{W}_p\ v_p^{\,2} \qquad (2)$$

Combining equation (1) and (2) gives the third equation of interest:

$$\text{Power} = \frac{v_p}{2g_0} \times \text{Thrust} = 1/2(I \times \text{Thrust}) \qquad (3)$$

From these equations, we see that required power (eq. (2)) goes up as the square of the propellant velocity, while thrust (eq. (1)) goes up only as the first power of propellant velocity. So we find (eq. (3)) that for a fixed power, the thrust must go down as propellant ejection velocity (or specific impulse) goes up.

Optimum Specific Impulse

This result can also be stated as follows: Although the propellant weight can be reduced by increasing the ejection velocity indefinitely, the electric power required for a given thrust goes up, and the required power-generating system thereby gets larger and heavier. What must be done is to find the propellant ejection speed (specific impulse) for which the <u>sum</u> of the propellant weight needed and the weight of the electric power generating equipment is minimized. This ejection velocity will yield the greatest total amount of energy per unit weight, and the least total weight for the mission. It turns out that, for the likely missions and powerplant weights of the future, propellant velocities of 1 to 10 million centimeters/sec are desired. These velocities correspond to specific impulses of about 1000 to 10,000 seconds, as compared to the 450 seconds typical of high-energy chemical rockets, and 900 seconds which appear possible with nuclear rockets.

Specific Weight

Obviously, the weight of the electric power generating system plays a crucial part in determining the extent to which the removal of previous restrictions on propellant velocity can be effectively utilized to increase mission capability. An increase in the useful payload carried is possible only if the additional propulsion system weight for electric rockets is less than the saving in propellant weight. Mission studies have shown that, for a manned expedition to Mars, there is a net gain over nuclear rockets in useful load-carrying ability if an electric rocket can be developed with weight less than about 10 kilograms for each kilowatt of jet power produced. This number is called the specific weight of the electric propulsion system and is one of its most important figures of merit. Detailed weight studies indicate that it should be possible to build electric propulsion systems with specific weights as low as 3 kilograms per kilowatt in the sizes (greater than 1000 kilowatts) needed for manned interplanetary missions. Possibly even lower values of specific weight may be attainable with novel types of power generators now only in the conceptual stage.

Need for Electric Propulsion

With such electric propulsion systems, a manned expedition to Mars, with provision for landing and exploration on Mars, would require an initial vehicle weight of only one-half to one million pounds in a low orbit around the earth. (A fully-loaded jet liner, by comparison, weighs about 320,000 pounds). A conceptual design of such a vehicle is shown in figure 1. It could be launched into orbit by two to four of the Saturn-5 launching rockets now under development. Therefore, if suitable electric propulsion systems can be developed, there would be no need to build launching vehicles larger than, at most, four times the size of Saturn-5. With some capability of orbital assembly, Saturn-5 would be quite adequate for launching manned interplanetary missions. The billions of dollars which would be required to develop larger launching vehicles and the extremely cumbersome facilities to handle them, could then be used for other purposes.

In addition to greatly reducing the magnitude and cost of the boosters for full-scale manned planetary expeditions, development of adequately light-weight and reliable electric propulsion systems would simplify the unmanned scientific probing of all regions of our solar system. This is illustrated in Table I, where the payload that can be carried to various destinations is compared for three space vehicles, one propelled by electric rockets (with two different specific weights), one by a nuclear rocket, and one by chemical rockets using hydrogen-oxygen propellant. The table shows that the nuclear and chemically-propelled vehicles, even though they are much heavier than the electrically propelled vehicle, could successfully carry out only part of the desirable solar-system exploration missions. The electric rocket vehicle has enough payload for all of the missions shown, and could be launched by the medium-sized Saturn-1B rocket now under development, while the nuclear and chemical-propelled vehicles would require a much larger booster. There is, therefore, a strong incentive to develop an electric propulsion system in the weight and power range indicated by Table I, so that the number and size of vehicles that must be developed to accomplish the desirable scientific probe missions thoughout the solar system can be reduced.

Development of Electric Propulsion Systems

What is the prognosis for successful development of electric propulsion systems in the sizes and weight ranges needed? As yet, no clear guarantee of success is possible. Most engineers and scientists in the business believe that success will ultimately be achieved, but the estimated times required vary from a few years to more than fifteen years. The major unresolved problems are in the development of suitable electric power generating systems.

The status of the other major component, the thrustor, which uses the electric power to eject the propellant, is fairly good. There is considerable confidence that thrustors with adequately high efficiency, low weight, long lifetime and reliability for interplanetary missions can be developed in the next few years, even though there is still a great deal to be done to improve all of these performance categories.

Status of Thrustors

The ultimate basis for this relatively high confidence in the success of thrustors is that a number of ion accelerators with characteristics similar to those needed for mission application have been conceived, designed, built, and extensively tested in vacuum chambers. Their operation, performance, and problems are consequently quite well understood. The most successful of the ion thrustors so far developed is one conceived and investigated at the NASA Lewis Research Center (reference 14). This thrustor (illustrated in figure 2) ionizes the propellant atoms by bombarding the propellant vapor with electrons in the ionization chamber. A weak magnetic field is provided in the ionization chamber to make the electrons spiral around on their way to the chamber wall, thereby increasing the probability of collision with propellant atoms. The resulting propellant ions are extracted from the chamber by means of an accelerating grid, to which an appropriate voltage is applied. A second electron emitter, in the ion beam, provides the electrons needed to neutralize the ion beam so that it does not break apart due to mutual repulsion of the like charges, and also so that the vehicle will not accumulate a net negative charge.

Other ion thrustors, using a technique called contact ionization (rather than electron bombardment ionization) have achieved comparable performance [illegible]. These thrustors use cesium as propellant, because cesium is the most easily-ionizable element. If cesium is passed through a hot porous tungsten material, the cesium atoms lose an electron to the hot tungsten surface, thereby becoming ionized and subject to electrostatic acceleration. Successful designs of this type are being developed for the Air Force and NASA by Electro-Optical Systems, Inc. and by Hughes Research Laboratories, respectively.

Brief space flight tests of these thrustors are planned for this year to verify that actual performance in space is the same as that obtained in laboratory vacuum chambers.

Although use of ion thrustors for primary propulsion depends on availability of appropriate large electric power systems, small units (using a few kilowatts of power) are under development to serve another function, namely, to maintain the position and orientation of various types of satellite (communication, weather, astronomical) for long periods of time. Ion thrustors are ideally suited for this function, because of their small thrust and very low propellant consumption rate.

Other electric thrustors are currently being studied which it is hoped will either cover the lower range of propellant ejection velocities (one to five million centimeter/sec) at higher efficiency than is possible with ion thrustors, or will increase the thrust attainable with a given thrustor size and weight. These other types consist of: 1) electrothermal thrustors, which use the electric power to heat the propellant, 2) plasma thrustors, which generate and accelerate an electrically-conducting gas, and 3) colloidal particle thrustors, which are like ion thrustors, but use charged particles that are much heavier than atomic ions.

The reason that atomic ion thrustors become less efficient in the lower range of ejection velocities is that a certain fixed amount of energy is required to strip an electron from the atom. If this non-useful energy is large compared with the useful energy (ejection velocity) then the efficiency of converting electric power into jet power is low. At the lower ejection velocities required for some missions (specific impulses of 1000 to 5000 seconds), the efficiency of ion thrustors is therefore rather low; whereas at higher ejection velocities, the useful energy becomes much larger than the ionization energy, and the efficiency improves.

At the lower exit velocities, the efficiency can also be improved by: 1) reducing the ionization energy required or 2) increasing the useful energy per charged particle, at a given velocity, by increasing its mass (colloidal-particle thrustor). Both approaches are being investigated. The electrothermal devices reduce ionization energy by not requiring ionization at all, but simply heating the propellant and forcing it out through a nozzle, in the usual chemical or nuclear rocket manner. This is possible only in the lowest interesting range of specific impulse, from 1000 to perhaps 2000 seconds. The reason that it is possible to add more heat to the propellant electrically than in a nuclear rocket is that the propellant can be heated directly by an electric arc discharge, which is hotter than the temperatures that can be tolerated by solid tubes or chamber walls.

Although the electrothermal thrustors do not require ionization, there are other power losses which limit the efficiency. Some of the energy goes into breaking up the propellant atoms and molecules, and some ionization is obtained even though it is not desired. Consequently, the efficiency of electrothermal thrustors, although useable, is not as high in this lower specific impulse range as the efficiency of ion thrustors in the higher specific impulse range.

Considerable development effort on electrothermal thrustors is being sponsored by NASA and the Air Force, and operational units appear to be attainable within a few years.

The other improvement possible over atomic-ion thrustors, namely production of more thrust for a given thrustor size or weight, may be achievable with the plasma thrustors. (The electro-thermal thrustors already have this high thrust-density feature in the lower specific impulse range). The limitation arises, in the ion thrustors, because of the limited ion current flow that is attainable. This current can be increased only by increasing the voltage or by reducing the distance between the accelerating electrode and the ion source. The voltage is approximately fixed by the desired ejection velocity (specific impulse), and the accelerator spacing cannot be reduced below a certain limit without encountering either electrical breakdown (sparking and arcing), or problems of accurate maintenance of small spacing in the face of warpage, thermal stresses, and other design problems. Present estimates indicate that a thrust of only about one-tenth of a pound (45 grams) is possible for each square foot of thrustor exit area at a specific impulse of 5000 seconds. Plasma thrustors, since they operate on the principle of moving an electrically-conducting but neutral gas are not subject to this limitation. At present, however, none of the proposed plasma thrustors has achieved high efficiencies in continuous operation, and much research is required before feasibility is demonstrated.

Status of Power Generation Systems

For use with electric thrustors to form a primary propulsion system for interplanetary missions, the electric power generation system must fulfill the following requirements:

1) Operate in space

2) Operate for continuous periods of a year or more

3) Have low specific weight, (preferably less than 10 kilograms per kilowatt)

4) Have total electric power output in the range of several hundred to several thousand kilowatts

The first requirement immediately poses two rather severe problems: a) there is no way to get rid of waste heat in space except to radiate it away, and b) there are micrometeoroids in space capable of puncturing thin, lightweight containing walls. Problem (a) is bothersome because one of the most straightforward ways to generate large amounts of electric power in space is with a nuclear turbo-electric system (as illustrated in figure 1). Such systems are now in successful operation on Earth, using high-pressure steam (generated and heated by a nuclear reactor) to drive a turbine, which in turn drives an electric generator. The steam leaving the turbine still has a great deal of excess heat which must be removed, and the steam must be recondensed into water. This heat is usually removed by using cooling water from a nearby river. In space, with no rivers or even cooling air available, the required heat disposal would be accomplished by passing the steam through a radiator, consisting of a large array of tubes. Radiation of heat at steam temperature, however, is not effective enough and would require much too large an array of tubing to achieve the specific weight goals. A much higher radiator temperature, in the vicinity of 1100° F or more, is needed to reduce the radiator size and weight enough to achieve specific weights near 5-10 kilograms per kilowatt. This means that, instead of using water as the working fluid in the power generation cycle, we must use something that vaporizes and recondenses at much higher temperatures. The only good prospective "fluids" turn out to be alkali metals, such as sodium and potassium, which melt and vaporize in about the right temperature range. These fluids, however, are very corrosive under some circumstances, so that in reducing the radiator size to reasonable values, some difficult problems are introduced in finding suitable containment materials.

Problem (b) is related to problem (a) because the radiator, due to its large area, is particularly vulnerable to puncture by micrometeoroids. The working fluid cannot be allowed to leak out, because only a limited amount of it can be carried along. Fortunately, recent data on the micrometeoroid hazard indicate that radiator tube thicknesses that are needed for reasons of strength may also be adequate to reduce the probability of puncture by micrometeoroids to a negligible value. If this is verified, then no additional weight in armor or increased thickness may be required.

Requirement number two, continuous operation for periods of a year or more, imposes, for the nuclear turbo-electric system, severe problems of reliability and durability with high-temperature systems containing corrosive materials and high-speed rotating machinery. The problem is perhaps less severe for manned than for unmanned missions, since the crew can presumably undertake minor repairs in case of non-catastrophic failures. However, the corrosion and wear problems must be solved before feasibility of such systems is clear.

It may be advisable to digress a bit to explain the need for such long-time continuous operation. Basically, it is required because the specific weight of electric propulsion systems, in terms of kilograms per jet kilowatt, is much higher than for chemical or nuclear rockets, in fact, more than 100 times as high. Thus, even though the total useful energy per unit weight is higher for electric propulsion, it must be released at a much slower rate to maintain reasonable propulsion system weight. This low rate of energy release, which is the jet power (equation (3)), means that the thrust that can be generated is very low, since jet velocity must of course be kept high. Consequently, whereas with chemical or nuclear rockets, the thrust is applied in relatively short, very high power bursts, it must be applied continuously for long periods of time at relatively low values with electric propulsion. Typically, the thrust is of the order of two thousand to ten thousand times lower than the weight of the vehicle to be propelled. This means that the thrust must be applied two to ten thousand times longer than with a system having thrust equal to vehicle weight. For a round-trip Mars mission, for example, with a total round-trip time of about 450 days, thrust would be applied continuously with electric propulsion for almost the entire trip, except during the Mars exploration period.

Thus, by decoupling the power source from the thrustor, the restrictions on propellant velocity and specific impulse are eliminated, but at the cost of increased specific weight and operating lifetime of the propulsion system.

At the present time, the only power generation system under development which is of possible importance for major electric propulsion missions is the SNAP-50 program, jointly sponsored by the Air Force and the Atomic Energy commission. This system, if successful in achieving its goals, could provide the electric propulsion power for the space-probe vehicle listed in Table I, and would consequently be extremely useful.

The SNAP-50 system is a nuclear turbo-electric system, with potassium as the working fluid for the turbine-radiator system. Liquid lithium, however, is to be used to transfer the heat from the nuclear reactor to the potassium. The two-loop system avoids the problems of actually boiling and vaporizing in the reactor itself, since lithium remains liquid at the highest temperatures of the cycle.

In addition to the SNAP-50 program, extensive research on high-temperature materials, corrosion, bearings and seals, and the multitude of other problems related to nuclear turbo-electric power generation are being sponsored by NASA, the Air Force, and the Atomic Energy Commission. These advanced technology programs should lead to power generation systems which meet the four requirements stated at the beginning of this section.

Other Power Generation Methods

In the meantime, a number of promising improvements and alternative methods of meeting the electric propulsion power requirement are being studied. One method, which is closely related to the liquid metal nuclear-electric system previously described, consists of using a non-reacting gas, such as argon or helium, instead of the liquid-vapor cycle. This method eliminates high temperature corrosion problems, but requires a larger radiator for the same maximum cycle temperature. The overall specific weight is therefore higher than for the liquid metal vapor cycle.

A promising method which eliminates all rotating equipment is the so-called magnetohydrodynamic (MHD) power generation method. In this method, the turbine and generator of the previous systems are replaced by a duct surrounded by an electromagnet. The hot vapor of the working fluid is somehow made electrically conducting and is forced through the duct and its magnetic field. The conductive vapor in essence serves the function of the solid conductors (copper wires) in an ordinary electric generator, and a voltage and current are produced in the vapor as in the copper wires. The resulting generated power is thus obtained with no moving parts in the system except the working fluid itself. After condensation in the radiator, it is recirculated through the reactor by an electromagnetic pump which, again, has no moving parts. If this method turns out to be successful, the problems of bearings and seals for high-temperature, high speed turbines and generators would be eliminated. Furthermore, it might be possible to operate at somewhat higher temperatures than with the turbo-generator, thereby reducing the specific weight.

Another method for eliminating rotating equipment consist of using direct conversion of the reactor heat into electricity. This can be accomplished by building thousands of thermionic cells into the reactor itself. A thermionic cell consists basically of a hot element and a cooler element. The hot element has the property that electrons, the fundamental particles of an electric current, are boiled off at the temperatures attainable in the reactor. These electrons have enough energy to travel against a small voltage to the cold element. The cell thereby produces an electric current at a small voltage (of the order of 1 volt). By connecting many of these elements together, a sizeable power can be generated.

Difficulties with nuclear thermionic power generation include the actual design of a reactor containing thousands of these cells, each with electrical connections, coolant flow over the cooler element, and very close spacing between elements. The problems appear formidable indeed. Mounting the thermionic cells outside the reactor, with heat carried to them by a liquid metal, is an alternative approach, but is not as promising with regard to possible specific weight reductions. The MHD system would probably be preferable, if it worked, because of its more rugged nature and sizeable voltage output.

The preceding systems all are modifications and possible improvements of the basic nuclear reactor thermodynamic cycle approach to power generation. One possible substitution is the use of solar power instead of a nuclear reactor in the same systems. There appears to be no real advantage in such an approach, in that the same high-temperature and other problems exist. The only benefit would be the elimination of the nuclear reactor with its harmful radiation and need for radiation shielding. Substitute problems would be the need for a very large, extremely light-weight solar collector, which would concentrate and focus the solar rays on a heater. Little, if any, improvement in specific weight appears likely from this approach.

There is, however, an attractive possibility in the use of solar energy that could eliminate the radiator, turbine, generator, pumps, working fluid, and heat exchangers, as well as the nuclear reactor and shielding. This is the possibility of using thin-film photovoltaic cells. Solar cells have been used extensively on a wide variety of satellite and space probes, but they have so far been much too heavy and expensive to be considered for the hundreds of kilowatts needed for electric propulsion. Recently, however, it has been found that solar cells can be made by coating thin solid layers of materials such as cadmium sulfide on a suitable thin substrate (reference 21). The resulting cell is flexible, quite rugged, and resistant to radiation damage. As in other photovoltaic cells, a voltage is generated simply by the action of light rays on the cell surface. Current weights of these cells are about 35 kilograms per kilowatt, which is still somewhat too high for major electric propulsion applications, but indications are that specific weights of 10 kilograms per kilowatt or less may be attainable. If so, the only real problem is that of mounting the cells on a suitable inflatable structure for unfurling after launch into space. The size will, of course, be quite large - of the order of 500 square feet per kilowatt of electric power produced - but this problem appears simple relative to those of the nuclear turbo-electric system.

Although most of the systems described appear to have potential to achieve specific weights of about 5 kilograms per kilowatt, none of them appear capable of reductions below about 2.5 kilograms per kilowatt. Only two systems have been suggested which, at least conceptually, can achieve specific weights below 1 kilogram per kilowatt. These are a radioisotope cell system (reference 23) and a variable-temperature dielectric system (reference 22). Both of these systems require thin film techniques. Both methods appear worthy of further investigation. Possibly other methods based on new uses of thin films will be conceived in the future.

Concluding Remark

The development of electric propulsion systems with sufficiently low specific weight and long lifetime for interplanetary missions is difficult, but is likely to yield to good engineering design, materials studies, and stubborn perseverance. The ultimate benefits in terms of man's ability to travel greater distances through space in less time and with vehicles of more reasonable initial weight and size would appear to fully warrant at least the magnitude of effort now being expended.

BIBLIOGRAPHY

1. Oberth, H., Wege zur Raumschiffarht, pp. 409-423; Munich and Berlin, R. Oldenbourg, 1929.

2. Shepard, L.R., and Cleaver, E.W., The Atomic Rocket, Jour. British Interplanetary Society: Pt. I, Vol. 7, p. 185, 1958; Pt. II, Vol. 7, p. 434, 1948; Pt. III, Vol. 8, p. 23, 1949; Pt. IV, Vol. 8, p. 59, 1949.

3. Stuhlinger, E., Electric Propulsion System for Space Ships with Nuclear Power Source, Jour. Astronautics; Pt. I, Vol. 2, p. 149, 1955; Pt. II, Vol. 3, p. 11, 1956; Pt. III, Vol. 3, p. 33, 1956.

4. Stuhlinger, E., The Flight Path of Electrically Propelled Space Ships, Jet Propulsion, Vol. 27, No. 4, pp. 410-414, April 1957.

5. Moeckel, W.E., Baldwin, L.V., English, R.E., Lubarsky, B., and Maslen, S., Satellite and Space Propulsion Systems, NASA TN D-285, June, 1960. (Unclassified version of material presented at NACA Flight Propulsion Conf., Nov. 22, 1957.)

6. Moeckel, W.E., Propulsion Methods in Astronautics, Int. Series on Aeronautical Sciences and Space Flight, Vol. II; New York, Pergamon Press, 1959. (Presented at First International Congress of the Aeronautical Sciences, Madrid, 1958.)

7. English, R.E., Slone, H.O., Bernatowicz, D.T., Davison, E.H., and Lieblein, S., A 20,000-Kilowatt Nuclear Turboelectric Power Supply for Manned Space Vehicles, NASA MEMO 2-20-59E, March, 1959.

8. Irving, J.H., and Blum, E.K., Comparative Performance of Ballistic and Low Thrust Vehicles for Flight to Mars, Vistas in Astronautics, Vol. II; New York, Pergamon Press, 1959.

9. Langmuir, D.B., Problems of Thrust Production by Electrostatic Fields. Second Annual AFOSR Astronautics Symposium, Denver, April 1958.

10. Forrester, A.T., and Speiser, R.C., Cesium Ion Motors, J. Astronautics, Oct., 1959.

11. Moeckel, W.E., Fast Interplanetary Missions with Low-Thrust Propulsion Systems, NASA TR R-79, 1960.

12. Dangle, E.E., and Lockwood, D.L., NASA Experimental Research with Ion Rockets, ARS Paper No. 1126-60, presented at ARS Semi-Annual Mtg., Los Angeles, May 9-12, 1960.

13. Brewer, G.R., Etter, J.E., and Anderson, J.R., Design and Performance of Small Ion Engines, ARS Paper No. 1125-60, presented at ARS Semi-Annual Mtg., Los Angeles, May 9-12, 1960.

14. Kaufman, H., An Ion Rocket with an Electron Bombardment Ion Source, NASA TN D-585, Jan. 1961.

15. Mickelsen, W.R., Electric Propulsion for Space Flight, Aerospace Engineering, Vol. 19, No. 11, Nov. 1960.

16. Proceedings of the NASA-University Conference on the Science and Technology of Space Exploration, Chicago, Ill., Nov. 1-3, 1962. Session O. Power for Spacecraft, Session P. Electric Propulsion.

17. Spencer, D.F., Jaffe, L.D., Lucas, J.W., Morrell, O.S., and Shafer, J.I., Nuclear Electric Spacecraft for Unmanned Planetary and Interplanetary Missions, JPL Tech. Rept. No. 32-281, April 25, 1962.

18. Beale, R.J., Speiser, E.W., and Wornak, J.R., The Electric Space Cruiser for High Energy Missions, AIAA Electric Propulsion Conf., Mar. 11-13,

19. Stuhlinger, E., and King, J.C., Concept for a Manned Mars Expedition with Electrically-Propelled Vehicles. ARS Electric Propulsion Conf., Mar. 14-16, 1962, Paper No. 2374-62.

20. Moeckel, W.E., Electric Propulsion Systems for Mars Missions. Proceedings of Symposium on Exploration of Mars, Denver, June 6-7, 1963.

21. Shirland, F.A., Griffin, T.A., and Kiersen, G.H., Thin Film CdS Front Wall Solar Cells, Preprint 2566-62, Space Power Systems Conf., Santa Monica (Calif.), Sept. 25-28, 1962.

22. Beam, B.H., An Exploratory Study of Thermoelectrostatic Power Generation for Space Flight Applications, NASA TN D-336, Oct. 1960.

23. Low, C.A., Jr. and Mickelsen, W.R., An Electrostatic Propulsion System with a Direct Nuclear Electrogenerator, Aerospace Engineering, Vol. 21, No. 12, Dec. 1962.

24. Mickelsen, W.R., Comparative Performance of Electrostatic Rocket Engines, IAS Paper No. 62-74, presented at 30th Annual IAS Meeting, New York, Jan. 22-24, 1962.

TABLE I. - PAYLOAD FOR SPACE PROBE MISSIONS

(From refs. 17 and 20)

Mission	Payload, pounds*			
	Electric propulsion 25,000 lb initial wt. 6250 lb powerplant wt.		Nuclear rocket 79,000 lb initial wt. "advanced" reactor	Chemical rocket 300,000 lb initial wt. H_2-O_2 propellant
	6 kg/kw	12 kg/kw		
Place payload in 500-mile Mars orbit	11,600 (250 days)	7600 (250 days)	18,500 (230 days)	30,000 (230 days)
Jupiter Flyby	12,200 (500 days)	8300 (500 days)	15,000 (500 days)	38,000 (700 days)
Send payload 30° out of ecliptic plane	8,700 (232 days)	4400 (232 days)	4,000 (232 days)	No mission
Pluto Flyby	6,300 (1100 days)	2000 (1100 days)	No mission	No mission
Place payload in 2000-mile Saturn orbit	5,200 (1000 days)	1100 (1000 days)	No mission	No mission
Place payload in 2000-mile Jupiter orbit	4,000 (900 days)	300 (900 days)	No mission	No mission

* Number in parentheses are trip times, and "no mission" indicates insufficient payload)

NASA
C-49478

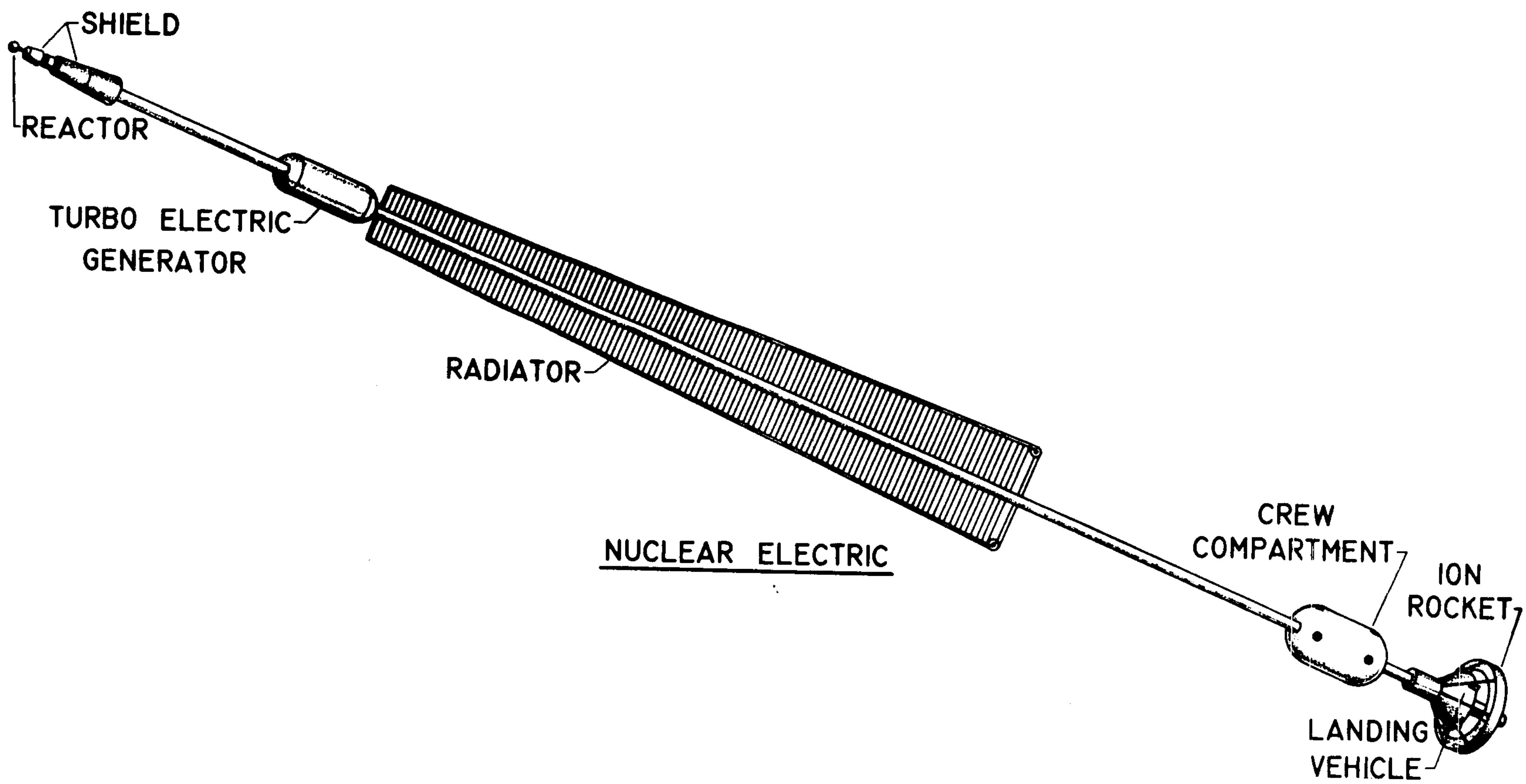

Figure 1. - Conceptual design of space vehicle for manned Mars mission.

Nuclear turbo-electric propulsion system.

NASA
CS-25138

ELECTRON-BOMBARDMENT ION ENGINE

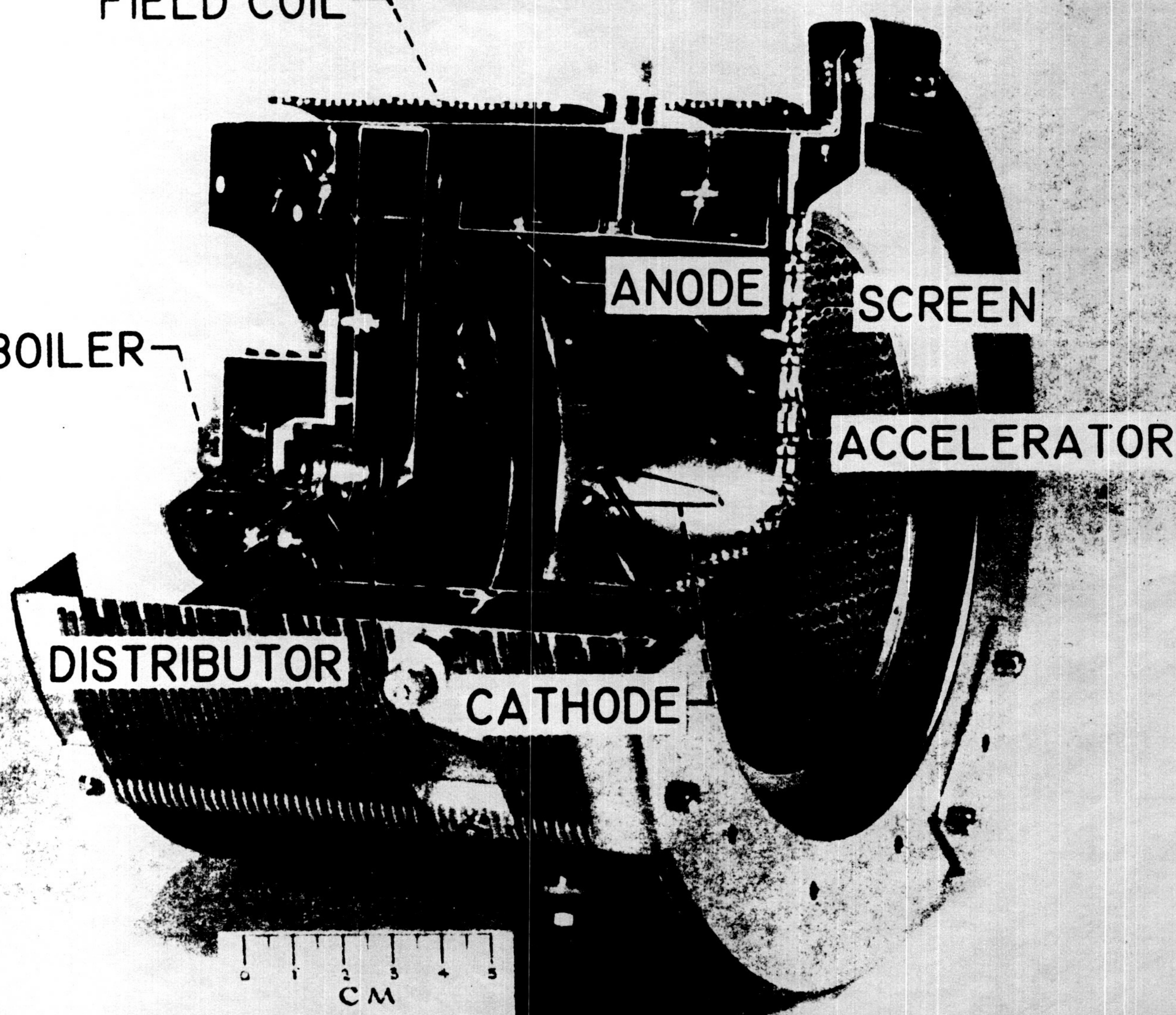

Figure 2. - Cutaway photograph of electron-bombardment ion thrustor. Using about 3 kilowatts, this unit produces a thrust of 7.5 grams at specific impulse of 5000 seconds.